

# **MINIATURE ASCENT VEHICLES DERIVED FROM THE NAVY'S AIR-LAUNCHED SATELLITE DEVELOPED IN 1958<sup>1</sup>**

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## **ABSTRACT**

This paper describes a planetary sample return launcher based on the PILOT microsatellite launcher developed by the U.S. Navy at China Lake in 1958. Developed in response to the launch of Sputnik 1, PILOT contained no moving parts (no gyros, no control surfaces, no thrust vector control) and only solid rocket motors. It was dropped from the wing of a fighter plane in a high-altitude climb and the first stage lofted it above the atmosphere. With slightly canted fins to make the rocket roll, it had a gyroscopic moment, but atmospheric drag acting on the fins kept the vehicle axis roughly tangent to the trajectory. At the apex a horizon sensor triggered the burn of two rocket stages which together achieved orbital velocity of 8500 m/s. A last stage was mounted backwards inside the payload. The spin of the payload kept it stabilized inertially in space, and thus the final stage was correctly oriented along the trajectory after half an orbit, where a timer fired it to give the "apogee kick" needed to put the satellite into a long-life orbit. This approach became known to this author since the project manager of PILOT happened to be this author's father.

## **INTRODUCTION**

Return of carefully-selected samples from planetary surfaces is one of the most important objectives of NASA planetary science and future human exploration programs. Analysis of these samples in the sophisticated laboratories of Earth will give vastly more scientific and engineering knowledge of those bodies than can be gained from any feasible near-term miniaturized in-situ instruments. Thus sample return is a high-priority objective for near-term planetary missions. However, sample return is perceived to be very expensive due to the very large mass multipliers that apply to the sample canister working backwards to the launch mass. What is described here is a potential sample return launch system which can be radically miniaturized compared to prior approaches. This system has no moving parts of any kind: no gyroscopes, no accelerometers, no control surfaces, and no thrust vector control. This concept is an enhancement and improvement of the formerly-classified PILOT microsatellite launching system

developed in 1958 by the U.S. Navy at China Lake, California at the [then] Naval Ordnance Test Station (NOTS).

## **PILOT**

Developed as a crash program in response to the Soviet launch of Sputnik 1, the PILOT (informally called NOTSnik) all-solid rocket launcher was about an order-of-magnitude lighter than conventional orbital launch vehicles developed before or since. It had no moving parts or control system. With a launch mass of under 1000 kg, it was dropped from the wing of a fighter plane climbing at about 70 degrees from horizontal and used a first stage to lob it above most of the atmosphere. It had fins which were slightly canted to make the rocket roll slowly about its axis, giving it a gyroscopic moment. The atmospheric drag acted on the fins to keep the vehicle axis roughly tangent to the trajectory. About 80 Km up, the trajectory became approximately horizontal. An optical detector sensed the horizon, triggering the second stage when the vehicle was approximately horizontal. A third stage was ignited by the burnout of the second stage. These latter two stages together achieved almost the entire Earth orbital velocity of 8500 m/s. A fifth stage was mounted backwards to the other stages at the geometric center of the annular payload, initiated by a 53 minute timer. The gyroscopic moment of this oblate payload kept the payload stabilized inertially in space for enough time to go half way around the Earth. The last stage, backwards to the flight direction at launch, kept its inertial direction and was thus pointed forward along the orbital direction on the other side of the orbit. When the timer fired the last stage it gave the "apogee kick" needed to put the satellite into a long-life orbit. A film describing the system was declassified in 1985, and represents most of what is known about the system, although it has been described in the non-technical media,<sup>1,2,3,4</sup> and the basic approach has been mentioned or briefly described in previous technical publications.<sup>5,6</sup> This approach became known to this author since the originator and project manager of PILOT happened to be this PI's father (deceased in 1994).

The film indicates that the concept originated very shortly after the launch in October 1957 of Sputnik 1. On 15 November 1957, the China Lake team briefed

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the Chiefs of the Bureau of Ordnance and Aeronautics of the U.S. Navy on the concept. Funding was received from the newly-formed Advanced Research Projects Agency (ARPA, now DARPA) in early March 1958.

The total vehicle has a length of about 15 feet and a weight of about 2000 lbs. The system was loaded onboard an F4D fighter aircraft in the manner of an external fuel tank. Fired by a lanyard and a 5-second timer, the first stage boosted the vehicle above the atmosphere using a cluster of four "Hotroc" motors developed at China Lake. These motors each generated 14,000 lbs. of thrust for 4.8 seconds using 330 lbs. of extruded, internal burning JPN propellant. These four motors are ignited in opposing pairs at an interval of 20 seconds as a means to reduce structural stress and drag losses.

The vehicle coasted upwards for about 100 seconds to an altitude of about 50 miles. The four large fins were tested to 1350 lb-ft of static bending moment and full aerodynamic loads while undergoing 25g acceleration using the China Lake supersonic test track. They were constructed of an aluminum frame machined from a single billet of material, covered with a stainless steel skin, and filled with foam. These fins were canted to produce a slow roll (reportedly about 300 RPM, although not stated in the film). This roll had to be high enough to spin-stabilize the upper stages but low enough to keep the vehicle approximately tangent to the trajectory at the apex of the ballistic arc of the first stage.

The next stage was a rocket motor developed at the Allegheny Ballistics Laboratory. It was initiated by a horizon-sensing telescope which looked out at a fixed angle of about 12 degrees from the vehicle spin axis. This approach to firing the orbit injection stages at the correct horizon angle helps reduce final injection errors to an acceptable level. The horizon sensing telescope was part of a shroud assembly which formed the launch rails for the upper stage system. The shroud assembly had a weight of about

40 lbs and was designed to withstand 40,000 lbs of axial load and bending moments of 10,000 lb-ft.

The next stage was a rocket motor developed at China Lake, which was about 18 inches long and 8 inches in diameter. It had a chamber pressure over 1000 psi using a wall thickness of only 0.025 inches. It produced over 1000 lbs of thrust for 5.7 seconds. At the end of this burn, the vehicle had achieved orbital injection velocity of about 19,000 miles per hour. All rocket stages were machined down to near the limits of withstanding the expected forces, and all the nozzles were sprayed with a ceramic coating to protect them from excessive degradation during each of the burns.

The final stage was a small 3-inch rocket motor containing 3/4 lb of X14 propellant. This rocket motor had a wet mass of 1.25 lbs and produced 172 lbs of thrust for 1 second. The payload was a 2.2 lb telemetering package formed in a torroid 8 inches in OD, 3 inches in ID, and 2 inches deep. It contained an FM transmitter with 3 subcarrier oscillators, accelerometers, temperature sensors, and a 30-hour battery, all potted in foam. The last stage was fired using a 3200-second timer so as to be approximately at apogee at the time of firing.

From this data given in the film, we can reconstruct the likely system characteristics of PILOT. These are shown in Table 1. Here the column "Stack Mass" refers to the mass of the stack at the end of the burn of that stage (e.g.  $M_{final}$  in the rocket equation). We have estimated that all the upper stages have the same specific impulse that we compute for the last stage. Based on the fact that it coasted only to an altitude of 50 miles, we conclude that almost 750 m/s of the first stage  $\Delta V$  was lost to drag in the lower atmosphere.

Some additional information exists beyond what was in the film, based primarily on a conversation on 29 July 1998 between this author and Dr. John Nicolaides, who had been the program manager at the Navy in Washington, D.C. at the time PILOT was developed. This conversation indicated that:

- Six vehicles were built and launched from the Western Missile Test Range off Point Mugu,

Stage Number	Propellant Mass (Kg)	Dry Mass (Kg)	Stack Mass (Kg)	Log of Mass Ratio	Specific Impulse (sec)	Delta-V (m/s)	Contribution to injection velocity
1	600	70	305.9	1.09	200	2128	Only horizontal component $\cos(70^\circ)$
2	204	17	31.9	2.00	230	4510	54%
3	11.25	1.7	3.6	1.41	230	3175	38%
4	0.34	1.6	1.6	0.19	230	434	Not included in injection velocity
Totals	815.59	90.3				8413	Total Orbit injection velocity

Table 1: Approximate system characteristics of PILOT based on data given in film

California in July and August of 1958.

- One vehicle failed due to an explosion of one of the four Hotroc boosters.
- One of the vehicles failed due to an instability which occurs as the increasing spin angular frequency passed through the natural frequency of the airframe in pitch or yaw.
- Two additional vehicles may have failed for unknown reasons.
- Two successful launches were reported by John Nicolaidis to senior advisors of President Eisenhower in July and August 1958, based on radio tracking data from the southern hemisphere. One of these successes was on the first attempt, where the pilot briefly lost control of the aircraft at the time of launch and in the confusion mistakenly reported that the launch vehicle had failed.
- The team at China Lake knew that there had been some indication of radio signals received in the southern hemisphere, but there was no claim of any successful orbital launch. The team was never advised that two of the launches were considered successful and reported as such to senior advisors to the President. The focus of the program was to counter possible threats to the Navy fleet posed by Soviet reconnaissance satellites, and so it was felt that any claim of success in orbiting the satellites would be counterproductive to this goal.
- This system evolved into the Caleb program and ultimately into the U.S. air-launched antisatellite system.
- No written reports on PILOT were created.

### MINIMAV

In May 1998, this author was involved in a small study team at JPL addressing ways to reduce the cost and complexity of Mars Sample Return. The leader of the team, Dr. Daniel McCleese, put forward the following question: "What is the smallest vehicle that can reach orbit from the Mars surface?" My father had shown me the PILOT film at the time he received a copy on video tape from China Lake after it was declassified in the 1980s, and we had discussed this question at length (for Earth, not for Mars). The premise of PILOT had been that the only practical limitation to miniaturization of a launch vehicle is the density of the atmosphere. The rocket equation gives the final velocity in terms of the exhaust velocity and the ratio of the initial mass to the final mass. Since only mass ratios are involved, the rocket equation gives no information about absolute masses. Were it not for the atmosphere, a miniature multi-stage rocket could achieve orbital velocity at a scale potentially

limited only by issues such as the thickness of combustion chamber walls in terms of atomic dimensions.

However, the existence of a significant atmosphere changes this conclusion. The drag losses on a vehicle are a strong function of the ballistic coefficient: unless the vehicle has a comparable mass per unit frontal area to the mass of the atmospheric column of the same cross-section, it will lose a great fraction of its kinetic energy passing through the atmosphere. This effect is aggravated since the burn time of a small rocket will be much shorter than that for a large launch vehicle, so the system will achieve orbital velocity very soon after launch. It will thus be low in the atmosphere at this high velocity, and so the drag losses will be very high. PILOT was launched from an altitude of about 11 km, where the mass density of an atmospheric column is equivalent to less than 2 meters of water. Since the density of rocket fuel is comparable to that of water, this means that the smallest vehicle which can reach orbit from that altitude is a few times 2 meters long. The PILOT launcher was about 4.5 meters long.

Mars, on the other hand, has an atmospheric density only equivalent to a water column of less than 10 centimeters. Thus we conclude that the smallest launch vehicle which can reach orbit from the Mars surface is perhaps only 25 cm long. Its payload would be too small to be of use for Mars Sample Return, but this conclusion gives encouragement that small sample return vehicles are possible.

We can form a conceptual design of a Mars Ascent Vehicle (MAV) along the system model of PILOT. Since the resulting vehicle will be radically smaller than previous MAV concepts (the baseline MAV in early 1998 had a mass of about 400 kg), it was dubbed MiniMAV. This concept was presented to the NASA Mars Exploration Architecture Workshop in Pasadena, CA in July, 1998. A reasonable MiniMAV has a total launch mass of about 20 kg, as given in Table 2. This system uses a series of identical stages to keep development cost down, even though these identical stages are not an optimal arrangement of a staged rocket. Even so, the system should be able to orbit a final assembly with a mass of almost 3 kg.

The identical stages used in this system would be custom developed, but with performance parameters which do not push the state of the art. Each stage has 3.75 kg of propellant with a specific impulse (ISP) of 280 seconds in vacuum. (The first stage has a lower effective ISP since it does not burn in vacuum.) The propellant mass fraction assumed for these stages is 90%, which is relatively easy to achieve for this size rocket motor. 210 grams is devoted to staging hardware, pyrotechnic cabling, etc. between the stages, and another 620 grams is devoted to fins on the first

	Stage 1	Stage 2	Stage 3	Injection Stage	Circular- ization	Trim Thrusters
Specific Impulse (sec)	260	280	280	266	240	240
Propellant mass fraction	0.90	0.90	0.90	0.71	0.60	0.50
Staging mass fraction	0.20	0.05	0.05	0.00	0.00	0.00
Effective propellant mass fraction	0.75	0.86	0.86	0.71	0.60	0.50
Fraction of insertion velocity	0.08	0.22	0.33	0.37	0.00	0.00
Delta-V of stage (m/s)	534.9	801.5	1222.6	1371.4	135.8	500.0
Mass at ignition (Kg)	19.81	14.81	10.43	6.06	3.08	2.90
Propellant mass (Kg)	3.75	3.75	3.75	2.97	0.18	0.69
Casing mass (Kg)	0.42	0.42	0.42	1.21	0.12	0.69
Staging mass (Kg)	0.83	0.21	0.21	0.00	0.00	0.00

Table 2. A MiniMAV system configuration with a launch mass under 20 kg.

stage. The final "injection" stage is assumed to contain the Mars sample and the small backwards circularization stage inserted from the front along the centerline of the injection stage (Fig 1). Some of the propellant is assumed offloaded from the nominal load of the injection stage to accommodate these modifications. The injection stage is carried as part of the sample canister following burnout, as is the circularization stage. The fuel mass penalty to the circularization stage for keeping the injection stage attached to the payload is smaller than the mass of any reasonable release system.

Since the Mars sample must be at the geometric center of the injection stage/sample canister (due to uncertainties in its mass properties), it is most easily accessed by removing the circularization thruster at the front of the assembly. This leads to the interesting possibility that this assembly can be covered with a frangible bioshield skin which is ripped off by the ignition of the injection stage. This bioshield can protect a nominally sterilized sample canister from contamination with possible Mars organisms. At the time of the burn of the circularization motor, the aperture and seam where the sample was inserted can be sterilized by the hot rocket exhaust. Thus many or all planetary protection concerns about the sample canister can be remedied by a simple approach.

A difficulty with the PILOT approach is that, in its original form, there is a very limited ability to precisely specify the orbital parameters which result from this unguided launcher. We have included "trim thrusters" around the beltline of the sample canister which can be fired by remote control from the rendezvous vehicle to correct for orbit injection errors. These trim thrusters will supply a combined  $\Delta V$  of 500 m/s, which is expected to be more than adequate to correct the final orbit of the sample canister. Several additional means have been

formulated to minimize the final orbit errors which are the subject of a preliminary patent filing by Caltech, and thus will not be addressed here.

The injection stage needs to have an oblate mass distribution to remain spin-stable during the half orbit prior to circularization. This is largely accomplished by placing the trim thrusters around the "equator" of the spinning assembly. Additional needed components (pyro firing circuits, radio beacon, solar cells, battery and other functions) can be arranged near the perimeter of the injection stage and spin balanced. Approximately 800 grams is available for these functions, assuming the sample mass is 200 grams.

A significant challenge to acceptance of this technology is the need for ultraminiature pyro initiators, cabling, and safe/arm/firing circuits. The NASA Standard Initiator and the standard pyro firing cable accepted by normal NASA safety procedures are much too massive to be acceptable for this approach. Fortunately, common practice for both military and commercial pyrotechnic systems include a full complement of components which are acceptable, but they would need to pass NASA safety qualification.

The Mars sample (and the circularization thruster) must be protected from the thermal pulse of the injection stage firing. Initial analysis indicates that this is not a difficult problem, but the needed insulation is part of the 790 excess grams allocated to the "casing mass" of the injection stage over the other stages.

This sample return rocket is small enough that it could be carried directly aboard a Mars rover. This would obviate the need for having the rover return to the lander to disgorge its samples and the need to have a manipulator arm on both the lander and the rover for sample transfer and contingency sampling. The elimination of the need to periodically return to the lander would greatly increase the effective range of the rover, and the elimination of surface rendezvous would

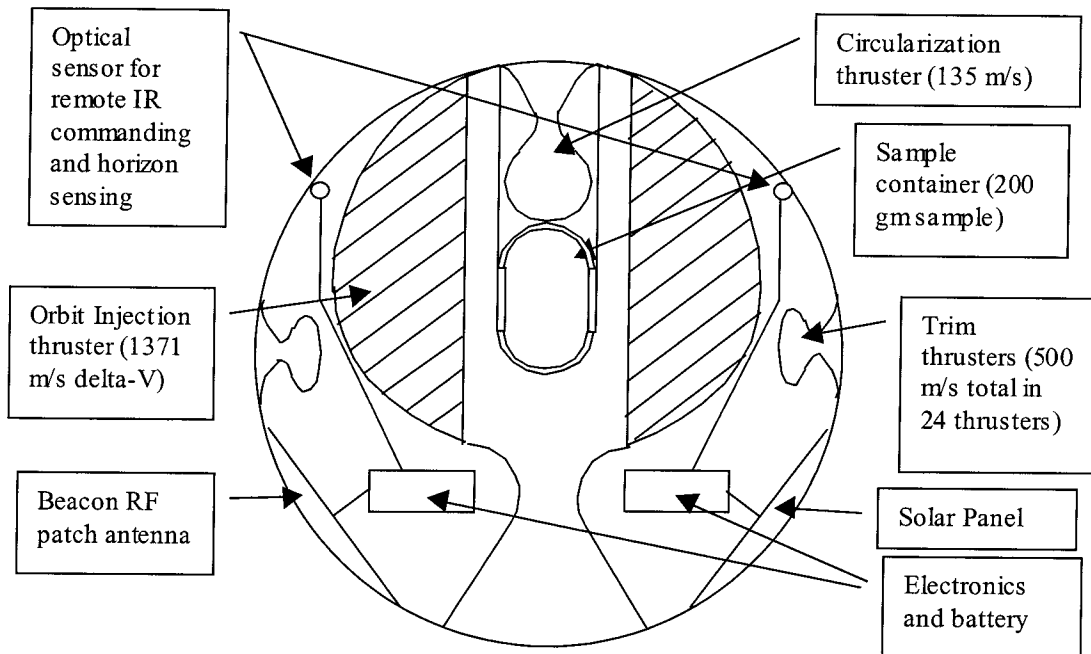


Figure 1. Conceptual layout of Sample Canister including injection and circularization stages

eliminate some significant technical challenges from the mission.

Coated with a surface which is white (for thermal reasons, if nothing else), the sample canister can be precisely located by optical means. An outer coating of Teflon(TM) is known to remain extremely white for long periods in Earth orbit, even with exposure to solar UV, solar wind and atomic oxygen. Extraneous coating by the pyrotechnic gases will be minimal due to the small quantities and high temperatures of that material. With a diameter of 25 centimeters, the sample canister will have 30 watts of solar power incident on it at the Mars distance. At a slant range of 200 Km, it has a brightness of about  $8 \times 10^8$  photons/m<sup>2</sup>sec. If it were sunlit against the dark side of Mars, which is illuminated with only 10 microwatts/m<sup>2</sup> of starlight, the signal-to-noise ratio is very large. For example, if a tracking camera has 500 pixels covering a 200 Km swath width (much larger than the range of possible canister orbits), then the signal to noise ratio is 190:1. Since only about 10,000 photons are needed to form a good image, a camera aperture of about 0.13 cm<sup>2</sup> is needed with a 1 second exposure time. With a standard CCD or CMOS image sensor, this requires an 8 mm F4.2 lens, which is small and inexpensive and easily incorporated in the rendezvous vehicle. A somewhat larger but still very practical camera could locate the

sample canister against the sky at ranges of thousands of km, even without a radio beacon.

An interesting possibility is to inject the sample container into a highly elliptical orbit. This will take a few kilos of extra propellant on the miniMAV, but it would make the issue of injection accuracy almost irrelevant and might reduce the wet mass of the rendezvous vehicle by perhaps hundreds of kilograms.

This concept is to inject at perhaps 4.5 km/sec or more instead of 3.6, and thus have an apoapsis at several or many Mars radii. This makes plane changes by the rendezvous vehicle relatively easy, since the orbital velocity at apoapsis is low. It would be important to put the rendezvous vehicle into an elliptical orbit such that the periapsis is over the miniMAV launch site. This would ensure that the line of apsides is the same for both the rendezvous vehicle and sample canister (since that line is expensive to modify, in terms of  $\Delta V$ ).

Acquiring the sample container would be more difficult for this case, but if periapsis is still around 300 Km and is arranged to be on the night side, then the same analysis about SNR over the dark face of Mars holds. If it were not convenient to have periapsis over the night side, there would still be a significant part of the orbit where it would be contrasted over the night side at relatively low altitude. And, as noted, it is not especially difficult to find the cansiter against the star field with a modest camera.

Since the sample canister would consist almost entirely of spent rocket casings interconnected by refractory foam (perhaps similar to Space Shuttle thermal protective tiles), this assembly could perhaps be used directly as the Earth Return Capsule. With a dry mass of 2.21 kg and a diameter of about 25 cm, the capsule would have a density about half that of water and a ballistic coefficient sufficiently low that it would dissipate its kinetic energy relatively high in the upper atmosphere. The thermal insulation which protects the sample from the heat pulse of the injection burn would similarly protect the sample from the heat pulse of entry. The relatively large surface to volume ratio and low thermal conductivity of the capsule implies that the capsule would quickly return to radiative equilibrium following the brief entry pulse. It will have a Reynold's number in the lower atmosphere of about one million and a corresponding drag coefficient of about 0.25, giving a terminal velocity about 52 m/s. If the capsule were to hit a very hard surface and crush about 5 cm upon impact, it would subject the sample to an acceleration pulse of about 2300 g's. Proper packaging of the sample should be possible so that no damage or breach of containment occurs with this level of acceleration.

A balloon launch test of this concept has been proposed, which would approximate the atmospheric density of launch from the surface of Mars. This suborbital flight would be tracked by radar to give performance data on each of the stages, and a sensor package in the sample compartment could validate the acceleration profiles and thermal pulse expected for the sample. Examination of the recovered canister would validate the performance of the planetary protection bioseal features, and give partial validation of the concept of using the canister as the Earth return capsule. A somewhat more ambitious balloon launch test, using additional Thiokol STAR 13 and 14 initial stages, has been proposed that could place the canister in Earth orbit so that further study of canister localization, orbit trim correction, rendezvous, Earth entry and recovery can be performed. This system would also serve as a flight validation of a system for balloon launch for a Venus sample return mission, since Venus and the Earth have very similar mass, radius, and upper atmospheric density profile.

### **CONCLUSIONS**

The fundamental concept of the PILOT system developed by the U.S. Navy at China Lake, CA in 1958 is that a properly configured spin-stabilized vehicle can achieve orbit without a control system, and that the degree to which it can be miniaturized is

limited primarily by the mass of the atmospheric column that the vehicle passes through. For planetary sample return missions, this can have profound consequences since the size of the sample return vehicle need not be limited by the technology available to miniaturize the avionics, sensors, and actuators. For Mars, where the atmospheric column is equivalent to less than 10 centimeters of water, this means that practical vehicles might be built with launch masses of less than 20 kg. For other planetary sample return missions where there is no atmosphere, such as from moons of the outer planets, Mercury, or the Earth's moon, the sample return vehicle might be miniaturized even more than this. With this approach even Venus sample return, despite its very thick atmosphere, is plausible with a launch vehicle mass of about 120 kg using a balloon launch.

### **ACKNOWLEDGEMENTS**

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